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RESEARCH MEMORANDUM

PRELIMINARY SURVEY OF POSSIBLE COOLING METHODS FOR HYPERSONIC AIRCRAFT

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SUMMARY

An investigation was conducted to determine the relative advantages and limitations of a number of fluids for use as either heat sinks or coolants for hypersonic aircraft. At flight speeds on the order of 18,000 feet per second it may be necessary to provide cooling for almost the entire aircraft structure. The cooling problem can be divided into two main categories: (1) high-level cooling in which the outside surface of the aircraft must be cooled to keep the surface from melting or oxidizing, and (2) low-level cooling in which the outside surface is allowed to reach equilibrium temperature but the internal support structure must be cooled to give it adequate strength. In general, cooling is much more difficult in the high-level regions.

Several heat sinks are probably feasible for hypersonic aircraft. Water appears best on a storage volume basis, and liquid hydrogen is best on a weight basis. Although lithium and lithium hydride have better heat-absorption capacities than either water or hydrogen, they do not appear to be suitable as heat sinks at the present time because of difficulties in handling molten lithium.

In the high-level regions suitable coolants would be hydrogen, water, sodium, or a sodium-potassium mixture. Surface cooling could possibly be eliminated in these regions by using uncooled surfaces made of silicon carbide, impregnated graphite, or tungsten with a silicide coating. In the low-level regions practically any coolant could be used for convection cooling, but water and hydrogen are particularly promising because they are also good heat sinks.

It may be possible to eliminate the necessity of taking useful volume from the aircraft for heat-sink storage by imbedding balsa saturated with water within the aircraft structure for low-level cooling and by using uncooled surfaces of high-temperature materials in the high-level regions.

INTRODUCTION

At hypersonic flight speeds aerodynamic heating may require that structural members be cooled in some way. This paper presents results of a preliminary study to determine feasible methods of cooling an aircraft structure at sustained flight speeds up to at least 18,000 feet per second at altitudes on the order of 180,000 feet. The aim of this study is to find reliable methods of cooling that are low in weight and volume, flexible enough to provide for variations in cooling requirements, not unduly complicated, and not hazardous to personnel.

A number of studies, such as references 1 to 6, have considered aircraft or aircraft-equipment cooling systems. Probably the most feasible method of cooling at hypersonic speeds within the Earth's atmosphere is to carry a heat sink or an expendable coolant within the aircraft. Air cannot be taken aboard for direct use as a coolant because of the extremely high temperatures the air attains from ram compression. Refrigeration systems do not appear practical, because they also require a heat sink at a reasonable temperature level, and the heat load to the sink is increased because of the inherent inefficiencies in the refrigeration system. It appears logical, therefore, to study the heat-transfer characteristics and heat capacities of a number of fluids for possible use as coolants and/or heat sinks.

At this stage of the investigation it is not expedient to make a design study of cooling for a particular aircraft, because configurations vary substantially. The configuration will affect the heat load, the volume available for the coolant system, and the problems associated with local hot spots. Instead, this study considers the problems associated with a range of heat fluxes that should encompass the range required for aircraft of the general class that would be capable of flight within the Earth's atmosphere at speeds up to at least 18,000 feet per second. Although much of the discussion is directed towards the cooling of wings, the same methods should be equally applicable to the fuselage.

THE HEAT LOAD

A detailed study of aerodynamic heating was not made for this paper, but some knowledge of the range of heat-transfer rates and equilibrium surface temperatures to be encountered in hypersonic aircraft is required in order to make an intelligent investigation of cooling methods. Heat-transfer rates and equilibrium temperature are influenced by speed, altitude, and aircraft configuration. The effects of speed and altitude on equilibrium temperature are shown in reference 1. It will generally be necessary to fly as high as possible at a given flight speed to reduce aerodynamic heating, but the altitude attainable is also a function of

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speed. As speed is increased, dynamic pressure is increased, and in addition the lift that must be developed by the wings is decreased owing to increasing centrifugal force. Both effects permit flight at higher altitudes. When satellite velocity (about 26,000 ft/sec) is reached, no wing lift is required. A study of the results in reference 1 reveals that aerodynamic heating for a glide missile is most severe for flight speeds between about 18,000 and 22,000 feet per second. This is the type of vehicle that would require cooling devices such as those studied herein.

The higher the wing loading, the more severe the aerodynamic heating will be, because attainable altitudes for a given flight speed will be lower. To determine an approximate idea of the magnitude of the heat fluxes (heat-transfer rate per unit area) and equilibrium temperatures to be encountered, heat-transfer calculations were made for a wing with a 78° sweep angle and a 0.25-inch leading-edge radius, flying at 18,000 feet per second at 180,000 feet with a wing loading of 20 pounds per square foot and at an angle of attack of 5°. The calculation methods of references 7 to 9 were used. The results of these calculations, which are presented in figure 1, are not to be considered authoritative but rather are estimates for a class of aircraft on which part of the present analysis is based. High heat-transfer rates are encountered for only the first inch of the leading edge, and beyond 6 inches on the lower surface and 1 inch on the upper surface (measured along the surface normal to the leading edge) the equilibrium temperature is less than 1800° F for a surface emissivity of 0.9.

There are several schools of thought concerning the best type of structure in the regions where the equilibrium temperature is 1800° F or less. Some of the possible types of construction are (1) building the structure of materials such as a molybdenum alloy that can safely withstand this temperature if a suitable oxidation-resistant coating can be found, (2) building the structure of high-temperature alloys such as those developed for gas-turbine engines and applying enough internal cooling to reduce the structure to a safe operating temperature (e.g., 1200° to 1600° F), and (3) utilizing essentially an uncooled outer skin, a layer of insulation, and an internally cooled support structure as illustrated by the low-level configuration in figure 2(b). In addition to the aforementioned components in the third type of construction, reinforcement channels such as illustrated in figure 2(b) would have to be placed periodically throughout the structure in order to fasten the outer skin to the cooled corrugated component and to add rigidity to the outer skin. The degree to which the internal structure would have to be cooled would be a function of the material used for the structure. Approximate temperatures for structures might be 250° F for aluminum, 500° F for titanium alloy, and 1200° F for a high-temperature alloy such as Inconel X.

The study herein will consider primarily the cooled structures. Only 7 inches of the wing (measured normal to the stagnation line) near the leading edge requires surface cooling. This will be called the "high-level" cooling region. In the regions where the equilibrium temperature is 1800° F or less ("low-level" cooling regions), only the interior structural members of the wing will require cooling. The temperature to which the structure must be cooled depends on the materials. The heat flux to these regions is quite low - less than 1/2 Btu/(sq ft)(sec) for a 1-inch-thick layer of low-conductivity insulation such as Thermoflex.

THE HEAT SINK

A variety of elements and compounds shown in reference 1 are capable of absorbing a large quantity of heat if vaporized, but usually the vaporization temperature is higher than the 1800° F that is considered to be the maximum temperature for the aircraft. Figure 3 shows the heat-absorption capacity of several elements and compounds that may be suitable for an aircraft heat sink. The heat-absorption capacity is considered to be zero at the coolant storage temperature that appears practicable for the aircraft. For helium or hydrogen this temperature is the boiling temperature at a pressure of 1 atmosphere. Approximately 100° F is the storage temperature for the other heat-sink materials. The gases helium and hydrogen obtain their heat capacity from a high specific heat and a large possible temperature change. Water and the light metals, sodium and lithium, obtain most of their heat capacity through vaporization, as shown by the vertical lines on the figure. The temperature at which this vaporization occurs is influenced by the pressure level. Vaporization is shown for two pressure levels for each of the light metals and water. The compounds methanol CH_3OH and lithium hydride LiH obtain most of their heat capacity through dissociation. The dissociation occurs over a range of temperature and is a function of pressure level (ref. 10). The rate of dissociation is also influenced by the concentration of the dissociation products. Complete dissociation would require separation of the products from the original compound. This might be quite difficult for methanol, since both the original compound and the products would be in the gaseous state.

Figure 3(a) shows that on a weight basis the potential heat capacity of water is lower than that of any of the other heat-sink materials considered. Since volume is also an important consideration in an aircraft, the heat capacity on a volume basis is shown in figure 3(b). Water has the highest density of all of the heat-sink materials considered; consequently, it shows up better on a volume basis than on a weight basis. At 1500° F the volume of water required to absorb a given quantity of heat is only about one-sixth of the volume required for helium and about one-fourth of that required for hydrogen, but the weight of hydrogen required is less than one-third that of water.

The liquid metals and lithium hydride show up very favorably in figure 3 based on heat-absorption capabilities, but there are a number of difficulties associated with handling liquid metals at high temperatures (refs. 11 and 12). Lithium hydride dissociates to molten lithium and hydrogen; therefore, its handling properties will be similar to those for lithium. Lithium readily absorbs nitrogen and forms lithium nitride, which is extremely difficult to remove. In the molten form lithium nitride might almost be called the universal solvent, as it will dissolve practically all known containers at high temperatures. Further research is required on suitable methods of handling molten lithium before serious consideration can be given to its use in hypersonic aircraft. Sodium does not present such a difficult problem. If a system is kept clean and, in particular, free of oxygen, sodium can be handled very well up to temperatures of at least 1200° F. Mass transfer may be a problem at higher temperatures where sodium vaporizes. For flights of short duration this may not be a problem, but oxygen leaking into a system could cause oxide plugging if the sodium has to flow through any small openings.

Helium and hydrogen (particularly hydrogen) appear quite promising as heat sinks if the volume required is not excessive for the aircraft in question. The storage of a liquefied gas in aircraft appears to be feasible. Figure 4 shows some of the details of a possible method of tank construction. The tank is formed of concentric shells of stainless steel with Styrofoam insulation between the shells. The insulation, which should be about 2 to 3 inches thick, has a density of about 2.5 pounds per cubic foot. The volume where this insulation is located could be evacuated to reduce heat transfer. The insulation serves as a separator between the two shells. As a result, the load due to the vacuum on the outer shell is quite low and the shell can be quite thin. A thickness of from 0.010 to 0.020 inch seems sufficient. The inner shell should be thick enough to withstand the internal pressure, which should probably be about 50 pounds per square inch absolute in order to provide sufficient pressure to circulate the gas if it is also used as the coolant.

With the proper operating procedure, carrying the hydrogen aboard the aircraft should not be a hazard to personnel. Purging and explosion problems associated with circulation of hydrogen throughout the coolant passages in the aircraft are believed to be easily controllable and will be discussed later.

Methanol shows a very good heat capacity on either a volume or weight basis (fig. 3). A nickel catalyst is required for the dissociation reaction, and the dissociation products are carbon monoxide and hydrogen gas. A further investigation of the dissociation reaction of methanol concerning rates, vapor pressures, and removal of dissociation products from the methanol gas would probably be warranted.

The gas or liquid used as the heat sink may or may not be used as the circulating coolant. For simplicity it would be desirable for the heat-sink fluid to serve also as the circulating coolant, because it would then be unnecessary to carry and store more than one fluid. This would also eliminate the complexity, weight, and volume occupied by a heat exchanger.

HEAT-TRANSFER CHARACTERISTICS OF SEVERAL COOLANTS

Configurations Investigated

If the entire structure of an aircraft is to be cooled, combining the heat-transfer and structural members of the airframe is desirable. One method of doing this is to use a corrugated structure similar to that shown in figure 2. This structure can be lightweight and rigid and at the same time provide cooling passages in the corrugations. For this study no attempt was made to optimize the configuration on either a heat-transfer or a structural basis. Probably some compromise will ultimately be required, because a single configuration is quite unlikely to be best for both uses. The configuration for this analysis was arbitrarily chosen. Studies that have been made on similar types of coolant-passage geometries for turbine blades (ref. 13) have shown that there is considerable freedom in choosing configurations; therefore, the configuration chosen is probably adequate at the present stage of investigation and will be suitable for comparing the relative merits of various coolants.

Figure 2 shows configurations for high-level, low-level, and leading-edge cooling. In the high-level and leading-edge regions (figs. 2(a) and (c), respectively), cooling is applied directly to the outside skin in order to reduce its temperature to 1800° F. For simplicity the configuration directly at the leading edge is similar to that for the high-level region behind the leading edge, and the dimensions of the corrugations (triangular passages) are such that they fit into a leading edge having a 0.25-inch radius.

For the low-level regions (fig. 2(b)), the equilibrium temperature of the skin is 1800° F or less; therefore, direct cooling of the skin is not required. The outer skin probably would not be sheet material but would be a honeycomb or corrugated structure to provide added rigidity. The temperature that the interior structure can withstand will be considerably less than 1800° F; therefore, structural cooling is still required. A 1-inch-thick insulation blanket of Thermoflex (thermal conductivity equal to 1.1 (Btu)(in.)/(sq ft)(hr)(°F)) would considerably reduce the quantity of heat transferred to the interior structure. The corrugation configuration was assumed to be the same as used in the high-level regions. The permissible temperature level at the corrugations would depend on the materials used in the structure. Three approximate average temperature levels were considered: 250° F for an aluminum structure, 500° F for a titanium structure, and 1200° F for a high-temperature-material structure such as Inconel X.

Assumptions and Conditions for Calculations

As previously pointed out, the fluids used for coolants need not be the same as those used for heat sinks, although many of the heat-sink fluids are also excellent coolants, and the over-all system can be simplified if one fluid is used for both purposes. The following coolants were considered for the high-level and low-level regions: Air (closed system, not taken aboard from atmosphere), helium, hydrogen, water, steam, and NaK (a mixture of sodium and potassium having a melting point as low as 12° F depending upon the relative amounts of sodium and potassium). Although sodium was considered as a heat-sink material, NaK was considered as the coolant because of its lower melting point. Its heat-transfer properties do not differ greatly from those of sodium, so that, as a first approximation, results presented for NaK are also applicable to sodium.

High-level and leading-edge cooling. - The calculations for high-level and leading-edge cooling were made over a range of heat fluxes from 1 to 150 Btu/(sq ft)(sec), which more than covers the range shown in figure 1. The general calculation procedures were as follows: With gaseous coolants and with a given heat flux and coolant inlet temperature (coolant temperatures from 0° to 1000° F were assumed), the coolant flow was set to give a wall temperature of 1500° F at the coolant inlet. Variations in wall temperature, coolant temperature, coolant pressure loss, coolant Mach number, and pumping power were calculated as a function of distance from the inlet using methods similar to those in reference 13 and using average heat-transfer coefficients for triangular passages from reference 14. The inlet pressure was assumed to be 50 pounds per square inch absolute.

When NaK was used as a coolant, the wall temperature was assumed to be the same as the coolant temperature because of the extremely high heat-transfer coefficients that are obtained with liquid-metal cooling. As mentioned previously, sodium (or NaK) can be handled easier if its temperature is kept at less than about 1200° F. In addition, NaK boils at about 1440° F at 1 atmosphere of pressure, so that the surface temperatures will have to be lower when NaK is used as a coolant than when the coolant is a gas. For these calculations the flow rate of NaK was set by allowing its temperature to rise from 1200° to 1350° F in 5 feet of coolant-passage length. The pressure loss and pumping power were calculated for these conditions.

With water as the coolant, the wall temperature was assumed to approach the water temperature of 281° F (saturation temperature at 50 lb/sq ft abs). The coolant-flow rate was set to allow 10 percent of the water to vaporize in 5 feet of coolant-passage length. The pressure loss and pumping power were calculated for these conditions.

Low-level cooling. - The heat fluxes used for the low-level-cooling calculations were determined by the amount of heat that would be conducted through the 1 inch of Thermoflex insulation for a surface temperature of 1800° F and the assumed internal structure temperature (a function of material used). For gases the flow rate was that which resulted in an inlet Mach number of 0.05 and the coolant temperature for an inlet wall temperature of the internal structure of 200°, 400°, or 1100° F for aluminum, titanium, or Inconel X structures, respectively. These assumed inlet temperatures are lower than the average temperatures previously mentioned, in order to allow for heating of both the coolant and the passage wall along the passage length. Variations in wall temperature, coolant temperature, coolant pressure loss, coolant Mach number, and pumping power were then calculated as a function of distance from the inlet. The inlet pressure was assumed to be 50 pounds per square inch absolute. 4783

Calculations using NaK as a coolant were similar to the high-level cooling calculations, except that wall temperatures and heat fluxes were lower. The NaK temperature was allowed to increase 100° F in 5 feet of passage length.

Calculations using water as the coolant were also similar to those for high-level cooling except that the heat flux was lower.

Comparison of Coolants

The question arises whether it is more advantageous to pass the coolant through all the passages in the corrugated structure or only through those passages adjacent to the outside surface. The cooling may be adequate with either procedure, but the pressure losses are less if the coolant passes through all passages because of the larger flow area. As a result, the comparison of coolants shown in figure 5 is for the coolant passing through all passages. Heat-transfer and pressure-loss results are shown for two heat fluxes. For NaK and water, however, heat fluxes will be higher than for the gaseous coolants for a given flight condition because of the necessity of reducing the skin to a lower temperature, as previously discussed. This lower temperature results in higher aerodynamic heat transfer to the skin and less radiation from the skin. The heat fluxes in figure 5 were therefore adjusted (increased) for NaK and water to make the comparison fair for all coolants.

Figure 5(a) compares the coolants for a heat flux of 150 Btu/(sq ft)(sec), which would be about the maximum that could be expected at the stagnation line of the wing. At a pressure level of 50 pounds per square inch absolute it was not possible to pass enough air or steam through the cooling passages to obtain adequate cooling, and consequently air and steam are not shown in figure 5(a). Calculations showed that it was not possible to cool with a heat flux of 100 Btu/(sq ft)(sec) with these two coolants either. With hydrogen and NaK the maximum length of passage that could be cooled was a little over 3 feet because of wall temperature rise with hydrogen and pressure-drop limitations with NaK. Helium could cool a little less than $1\frac{1}{2}$ feet of passage before a pressure-drop limit was reached.

If higher inlet pressures had been used, helium and NaK would have been better coolants, because they were limited by pressure loss and/or Mach number before they were limited by excessive temperature rise of the wall along the passage length. It was also interesting to find that, when helium and hydrogen were used as coolants, the wall temperature 1 foot from the inlet was actually less than the inlet wall temperature because of a transition from laminar to turbulent flow, which improved the heat transfer. This wall temperature reduction, which was small, is not shown by the figure. The inlet coolant temperature for hydrogen and helium was 0° F for the results shown. Water, which was allowed to vaporize partially, was the only coolant with enough heat capacity per unit volume to permit cooling a passage 5 feet long.

The power requirements shown are those required to restore the coolant to 50 pounds per square inch absolute at the end of the passage length considered and for the weight flow required for cooling a surface 1 foot wide. By restoring the pressure the coolant could be circulated again. This would result in better utilization of the heat capacity if the coolant were also the heat sink. If the coolant were not the heat sink, the coolant system would be a closed cycle with heat rejected to a heat sink by means of a heat exchanger. In this case the coolant would be recirculated after rejecting heat to the heat sink, and the pressure would definitely have to be restored. Because of the high pumping power requirements and the complication of installing a gas pumping system, the coolants hydrogen and helium, which are also heat sinks, would probably be exhausted at a low pressure and would not be pumped back to a high pressure for further circulation to better utilize the heat capacity. For the liquid coolants such as NaK and water, which would require recirculation, the power requirements are quite low.

From figure 1 it can be seen that within an inch of the leading edge the heat flux has dropped to a value of less than 10 Btu/(sq ft)(sec), so that only a very small portion of the aircraft will have the difficult cooling problem considered in figure 5(a). In figure 5(b) the heat-transfer and pressure characteristics of the coolants are shown for a heat flux of 10 Btu/(sq ft)(sec). The temperature rises, pressure drops, and coolant Mach numbers required for adequate cooling are probably satisfactory for passage lengths up to at least 5 feet for all coolants except air. With air as the coolant, wall temperatures may become excessive because of rapid coolant temperature rise. At this heat flux the flow in the coolant passages was found to be laminar for all coolants. For laminar flow the heat-transfer coefficient is essentially constant regardless of flow rate within the passages. As a result the wall temperature is controlled by controlling the coolant temperature. The results shown in figure 5(b) are for an inlet Mach number of 0.1 and an inlet coolant temperature on the order of 200° F less than the wall temperature. If very cold coolant were used, the structure would be overcooled, regardless of the flow rate, near the entrance of the passages. This overcooling can result in a waste of heat-sink capacity and in possible thermal stress problems. A possible method of at least partially overcoming this problem will be discussed later.

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LEADING-EDGE AND HIGH-LEVEL COOLING

Suitable High-Level Coolants

From the study of heat-transfer characteristics of the coolants considered in this investigation, it was found that air or steam would be almost completely unsatisfactory for heat fluxes of 100 Btu/(sq ft)(sec) or higher. Water is the best coolant if 10 percent of the water is allowed to vaporize along the length of the coolant passage. The pressure drop and pumping power for water are almost negligible, and quite long passages could be cooled. The next best coolant is NaK, followed by hydrogen and helium.

Freezing could be a serious problem with water or NaK as the coolant. Freezing could be encountered on the ground or at high altitudes before reaching speeds where aerodynamic heating becomes a problem. Possible solutions to the freezing problem are circulating the coolant in a heated state during freezing conditions, or adding propylene glycol to water to provide protection to approximately -65° F. Absolute protection against freezing would be required, because local blockage by frozen coolant could quickly result in overheating or burnout in some other portion of the system during hypersonic flight.

The freezing problem could be overcome by using hydrogen as the coolant in the high-level regions. With hydrogen, the cooling passage lengths would have to be chosen so that there were no limitations due to pressure loss or excessive wall temperature rise. For a heat flux of 150 Btu/(sq ft)(sec), a length of about 3 feet appears to be practical for a supply pressure of 50 pounds per square inch absolute. Higher supply pressures, which would probably require use of a liquid-hydrogen pump, would help a hydrogen-cooled system, both by permitting higher pressure drops and by causing a higher heat capacity per unit volume of hydrogen.

Another difficulty encountered with hydrogen as a coolant and heat sink is the utilization of the full heat capacity - that is, heating the hydrogen to a temperature near the wall temperature. Using the full heat capacity would require pumping so that the hydrogen could be recirculated, and the power requirements for this might be exorbitant. It would probably be more economical to discharge the hydrogen without completely using its heat capacity rather than to pump it. For the major portion of the high-level region where the heat flux is less than 10 Btu/(sq ft)(sec), quite long coolant passages could be utilized, but here again difficulty would be encountered in utilizing the full heat capacity of the coolant without pumping.

Operating Procedure with Hydrogen Coolant

One of the principal factors affecting the use of hydrogen as a coolant is the safety problem. Aside from the fact that hydrogen has very wide flammability limits, it is probably no more dangerous than the natural gas used for heating houses. By taking a few precautions it should be possible to circulate hydrogen through the aircraft structure without creating a hazard to the pilot.

One problem that must be overcome is the possible occurrence of a damaging explosion when hydrogen and air become mixed within the coolant passages of the aircraft structure. For most applications with hydrogen, the passages are purged with an inert gas such as helium before the hydrogen is introduced and after it is shut off. In the cooled aircraft structure, purging with helium may not be feasible because of complication, weight, and volume of the purging equipment. If the pressure within the structure were sufficiently low, however, detonation pressures would be tolerable, if a detonation did occur. In addition, the probability of a detonation is quite low for structure temperatures less than 750° F at the time the hydrogen and air are mixed (refs. 15 to 18). Detonation pressure ratios are a maximum of about 60 (ref. 19). At the altitudes that the aircraft will fly, the base pressures can be low enough that if detonation does occur the detonation pressure within the coolant passages will be 2 atmospheres or less. Since these passages should probably be designed to stand over 3 atmospheres, the occurrence of detonation would not damage the structure.

Figure 6 shows minimum altitudes as a function of flight Mach number where it would be safe to charge or bleed the hydrogen cooling system without danger to the structure if a detonation occurred. The figure is based on bleeding the system to the base pressure that would occur at the rear of the fuselage or at the trailing edge of the wing. The data for the figure are from unpublished experimental results of Reshotko and Cortwright at the NACA Lewis laboratory. As long as the hydrogen coolant is turned on or off and purged with air at altitudes above the minimum shown on the curves, there should be no hazard from a possible detonation. Flight above the minimum altitudes shown in figure 6 for corresponding Mach numbers seems feasible, and structural cooling would not be required at these conditions. These conditions should, therefore, be safe for turning the hydrogen flow on or off.

Utilization of Hydrogen Heat Capacity

Figure 3(a) showed that on a weight basis hydrogen is a superior heat sink at almost any temperature level, but it would be desirable to raise the temperature of hydrogen as high as possible in order to utilize its full heat capacity. This requires the use of hydrogen over a temperature range from its boiling point of about 40° R (-420° F) up to as high a temperature as possible. Two problems are immediately obvious: First, unless the proper design is used, some areas of the structure are liable to be much overcooled because of the very low hydrogen temperature as it comes from the storage tank. This overcooling wastes heat-sink capacity and may cause structural problems resulting from thermal expansions and contractions. Second, full utilization of the hydrogen heat capacity probably is not possible unless pumping is used. In the low-level region the low structure temperatures limit the temperature to which hydrogen can be heated; and in the high-level region pressure-drop limitations are a problem, as previously discussed.

There are at least two possible solutions to the problem of overcooling. One solution is the use of counterflow cooling of the corrugated structure as illustrated in figure 7. The metal used for the corrugations forms a natural divider, so that counterflow can be easily obtained if coolant is introduced to the passages above the corrugated divider on one end and to the passages below the corrugated divider on the other end. This method of cooling should reduce metal temperature variations along the length of the cooling passages and eliminate drastic overcooling. The highest metal temperature will be near the midpoint of the passage. Overheating could occur at this location if the passages were too long.

A second solution to the overcooling problem is the use of a closed-circuit cooling system and a heat exchanger to reject heat to the heat sink. The coolant in the closed circuit would have to be either hydrogen or helium to preclude freezing in the heat exchanger. The temperature of the coolant in the closed circuit could be high enough, however, that overcooling of the structure would not occur. A serious disadvantage of this type of system is the weight and complication of the heat exchanger and pump required for circulating the secondary coolant. Therefore, counterflow cooling is probably a better solution.

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This discussion of heat-capacity utilization is applicable to all materials that must be heated to high temperatures to exploit their full heat capacity. Further complications result if the material must be vaporized in order to obtain the major portion of the heat capacity. In these cases mixtures of both liquid and vapor must be handled, and under some conditions careful design is required to avoid "slugging" or burnout from vapor pockets.

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Upper Wing Surface as a Heat Sink

Figure 1 shows that the equilibrium temperature of the upper wing surface is less than 1000° F at distances from the leading edge greater than about 30 inches. This area could possibly be used as a heat sink for part of the high-level cooling region. Assuming that the coolant is a gas, such as helium or hydrogen, which would have a temperature intermediate between the high-level region and the heat-sink surface temperature, it might be possible to operate with a heat-sink surface temperature of about 1500° F and radiate heat picked up in the high-level region to the atmosphere. Calculations indicate that, if the area shown in figure 8(a) were used as a heat sink, between 30 and 50 percent (depending on surface emissivity) of the heat picked up from the wing high-level regions could be dissipated in the heat sink for an aircraft about 75 feet long.

Circulation of the coolant from the high-level regions to the heat sink requires additional ducting, a pump capable of operation at about 1650° F, a power source for the pump, and a modification to the wing structure to provide an additional corrugated structure on the outside surface for the radiating heat exchanger, as shown in figure 8(b). Whether this added complication and weight increase would offset the savings in weight in the fluid heat sink is not known. It appears doubtful, however, that use of the upper wing surface as a heat sink for high-level cooling would be warranted.

Effect of Coatings on Heat Flux

In areas where there is a high heat flux, large temperature reductions across relatively thin coatings are possible if the coating has a sufficiently low thermal conductivity. Stabilized zirconia coatings (Rokide "Z") are capable of withstanding very high temperatures. Calculations were made to determine the effectiveness of such coatings in the high-level regions. The results are shown in figure 9 for coating thicknesses of 0.025 and 0.050 inch from a metal temperature under the coating of 1800° F. The figure shows that temperature drops up to 1600° F are possible through the coating. The resulting higher surface temperature on the outside of the coating would reduce the convection heat transfer

and increase the heat loss by radiation for a given surface emissivity. The emissivity of zirconia coatings is only about 0.2 at high temperatures; therefore the coatings are not as effective in radiating heat as other common materials. The high-level heat fluxes that must be removed by cooling, shown in figure 9, are actually greater with the coating than without the coating for either thickness considered except for about the first 0.4 inch of the leading edge. It therefore appears that coatings will not effectively reduce high-level heat flux unless coatings can be found that have both low thermal conductivity and high emissivity. Such coatings capable of withstanding the very high temperatures required 4783
are unknown to the authors.

Transpiration Cooling

It is well known that transpiration cooling is theoretically the most effective method of cooling known at the present time. The analytical study of reference 20 shows how much coolant could be saved if transpiration cooling were used rather than convection cooling. The savings were appreciable with air as a coolant, but considerably smaller with helium as coolant. Experience and calculations (refs. 21 and 22) have shown that attaining the ideal coolant flows for transpiration cooling is extremely difficult, particularly if there are pressure gradients on the surface. A relatively complex metering device for the coolant will be required in order to obtain the proper flows at various locations for the range of altitudes that must be encountered, and the permeability of the surface must be controlled very carefully - more carefully than has been found practicable up to the present time. If these factors are not all carefully controlled, coolant flows can become greater than required for convection cooling, and local areas can overheat. Past experience in attempting to transpiration-cool turbine blades, which have many of the same cooling problems as wing leading edges, indicates that it would not be wise to attempt to use transpiration cooling in the high-level region at the present state of development.

Uncooled Materials

Figure 1 shows that, for a surface emissivity of 0.9, which is possible for several materials, the equilibrium temperature is 3400° F at the stagnation line and drops rapidly farther back along the leading edge. Conduction would probably lower the equilibrium temperature at the stagnation line. Several materials could probably be used without cooling at these temperature levels. Possible materials are tungsten, molybdenum, graphite, and silicon carbide. Reference 23 discusses research on coatings for molybdenum and tungsten. Information on graphite is given in reference 24. The properties of these materials are summarized in the following table:

Material	Coating	Maximum temperature, °F	Specific gravity	Tensile strength at 3500° F, psi	Surface emissivity
Molybdenum	$MoSi_2$	Coating, 3200 Mo melts, 4750	10.2	2,800	?
Tungsten	Silicide	Coating, 3800 W melts, 6150	10 to 19.3	10,000	?
Graphite	SiC	Coating, 2900 to 3300 Impregnated, 4000 C sublimes, 6600	1.6 to 1.7	4,000	0.9
Silicon carbide	None	Oxidizes, 2900 to 3360 Decomposes, 4500	3.2	12,000 at 2730° F	0.9

The materials that appear most promising in this table are tungsten, graphite, and silicon carbide. The surface emissivity of coated tungsten is not known by the authors. If it is low, the equilibrium temperature at the leading edge can be considerably higher than 3400° F. The specific gravity of tungsten is a function of sintering and pressing techniques. Graphite has excellent thermal shock properties; its strength increases with temperature up to about 5000° F; and its thermal conductivity is high - up to 5 times that of steel at room temperature, but decreasing with increasing temperature. Conduction would undoubtedly lower the stagnation-line equilibrium temperature for a graphite leading edge. The silicon carbide coating for graphite starts oxidizing at about 2900° F. The rate of decomposition is relatively slow at higher temperatures, but if the coating is thin the protection may soon be lost. Reference 25 states that graphite impregnated with refractory metal resists oxidizing atmospheres at temperatures well over 4000° F without dimensional change.

Recent developments with silicon carbide have made it a very promising material for use on uncooled leading edges. When about 7 percent molybdenum disilicide is added to almost pure silicon carbide, very good oxidation resistance has been reported at 3270° F and fair resistance at 3360° F. Pure silicon carbide starts oxidizing at about 2900° F, and oxidation is quite severe at 3300° F. The material has good thermal shock properties and a thermal conductivity about half that of graphite, and its strength at elevated temperature should be adequate.

Coatings for molybdenum have not been very satisfactory since, if a pin hole develops, the oxide of molybdenum, which is a vapor, will escape and the structure will disintegrate. In addition, the material has less high-temperature strength than tungsten, graphite, or silicon carbide, and the coating is good only to 3200° F; therefore, molybdenum is not considered a suitable uncooled material for the aircraft.

The use of an "uncooled" leading edge still does not entirely eliminate the cooling problem in this region of the aircraft. This uncooled member will have to be attached to a lower-temperature cooled structural member and will transfer heat by conduction at the points of attachment and by radiation in other areas. This radiant heat transfer from the uncooled member to the cooled members can be much less than 10 percent of the heat that would be transferred by convection to a cooled leading edge. In order to obtain low heat-transfer rates, low-emissivity coatings such as stabilized zirconia (Rokide "Z") should be used on the cooled portion of the structure behind the uncooled skin.

LOW-LEVEL COOLING

Several cooling methods are probably feasible in the low-level region of the aircraft. For a double-wall insulated structure the cooling could be by convection or by a built-in heat sink as shown in figure 10. A convection-cooled structure would have the advantages that the coolant could be turned on only when needed, thereby saving in the quantity of coolant required, and that the structure could be cooled to almost any desired temperature. It has the disadvantages that additional ducting would probably be required and a pumping device may be necessary for circulating the coolant if the tank pressure is not sufficient. In addition, lightweight structures that may be adequate for carrying the structural loads may not be suitable for very high internal coolant pressures. Coolant sealing could also be a problem.

The built-in heat sink is advantageous in that it is simple, that no tanks are required for coolant storage, and that more uniform cooling probably results. Its disadvantages are that renewing or replacing the heat sink between flights may be a problem, that there is less control over the temperature to which the structure can be cooled, and that the cooling cannot be turned off or on at will. A further possible difficulty with water as the heat sink is that the internal structure will be cooled more than it has to be because the low pressures encountered at altitude will result in boiling temperatures less than 100° F unless some pressurization is used within the wing. Quite possibly the increase in structural weight to permit pressurization would much more than offset the small weight increase in water required for overcooling.

Reducing the quantity of heat-sink fluid required with the built-in heat sink might be possible if the structure is capable of withstanding a higher temperature than the vaporizing temperature of the heat sink. In this case the quantity of fluid in the heat sink could be controlled so that it would provide adequate cooling in the early, high-speed part of the flight. In the later portion of the flight, when the flight speed and equilibrium temperatures have decreased, cooling may no longer be needed. By proper planning, the heat-sink fluid can be expended by that time with a saving in the quantity needed.

Other cooling methods could be used with single-wall uninsulated structures. For these structures, the material used in the structure should be capable of withstanding as high temperature as possible to reduce the heat load. Some convection cooling, probably with a gas because of the high temperature levels, could be used for controlling structure temperature. This would be similar to the method used for high-level cooling. Another possibility would be the use of full-depth honeycombs extending from lower to upper wing surfaces. Radiation from the lower surface to the much cooler upper surface would permit the use of uncooled structures over much of the wing surface if the structure could withstand temperatures up to about 1600° F. This temperature appears feasible for some applications where the structure loads are small if the structure is made of a nickel- or cobalt-base high-temperature alloy.

Convection Cooling

With the type of structure shown in figure 10 there are no real problems in heat removal by convection cooling with any of the coolants considered in the low-level regions. In general, calculations showed pressure losses and coolant velocities can be quite low. The permissible length of coolant passages and the coolant temperature rises can be controlled to a considerable extent by the mass-flow rates. Generally, the structure temperature will be very close to the coolant temperature. As mentioned previously, however, the utilization of heat-sink capacity is a problem with low-level cooling, because the heat sink cannot be heated to a higher temperature than the permissible structure temperature in the low-level region. In addition, the fact that the wall temperature will be very close to the coolant temperature can result in drastic overcooling in some areas, particularly if hydrogen or helium is used as the coolant. This overcooling can be avoided by using counterflow cooling, as discussed previously and illustrated in figure 7.

Because of the difficulty in fully utilizing heat sinks such as hydrogen, helium, or the liquid metals, water is a promising heat sink for the low-level regions. Water could be used for convection cooling by circulating the water through the structure at approximately boiling temperature at a rate that causes about 10 percent of the water to vaporize through the circuit. The vapor would be bled off and the remaining water would be recirculated.

A coolant system that circulates a liquid coolant suffers the disadvantage of considerable dead weight from the residual coolant that is necessary to fill the system. Since the system must remain full to the
end of the time that cooling is required in order to ensure circulation, 4783
this residual coolant cannot be used for a heat sink. Since a large area must be cooled, this can amount to almost as much weight in some cases as the weight of water that would be required for a heat sink for all the low-level regions on the airplane. If water is used as the heat sink, the complication of tanks and pumps for recirculating the water can possibly be eliminated by storing the water at the location where the heat is to be absorbed by the built-in heat sink, as will be discussed later.

The discussion up to this point on convection cooling in the low-level region has dealt with a double-wall insulated structure as shown in figure 10. For an uninsulated structure the cooling problem would be no worse than that illustrated in figure 5(b). The structure could be cooled with any of the coolants considered. Water or NaK would probably cool the structure more than necessary, and the total heat loads would be high. A gaseous coolant is therefore indicated. The best coolant would be hydrogen, and next best would be helium. Although the heat loads would be higher for the single-wall than for the double-wall construction because of higher permissible surface temperatures in the double-wall construction, the heat capacity of a heat sink like hydrogen could be more fully utilized because it could be heated to a higher temperature. As a result, the weight of heat-sink fluid may not be any higher for the single-wall construction. To determine which type of configuration is the best for cooling would require a heat-transfer analysis based on the structural configurations considered for a particular application and for the desired coolant and heat-sink fluid.

Cooling with Built-In Heat Sink

A heat-sink cooling system (fig. 10(b)) should be able to maintain a reasonably constant temperature by utilizing the heat of vaporization of a liquid. The temperature level can be controlled by choice of the heat-sink fluid and by the pressure level maintained in the system. For the fluids that were considered in this analysis, water would be suitable for temperatures up to about 212° F, sodium for temperatures from about 1000° to 1600° F, and lithium at temperatures in excess of 1600° F. A further requirement for heat-sink cooling is that the heat-sink material be capable of proper distribution over the areas that must be cooled, and that its distribution not be adversely affected by acceleration forces that might be experienced in the airplane. With all factors considered, water appears the most practical fluid for heat-sink cooling at this time.

The next question is how to hold the water in the proper locations under all flight conditions. Probably the most practical way of storing the water is to let it be absorbed in some porous material that can be distributed throughout the airframe. Most materials that are capable of absorbing large quantities of water will lose most of the water under acceleration forces of a few g's. An exception, however, is saturated balsa. Experiments conducted at the NACA Lewis laboratory have shown that balsa wood can be saturated to a specific gravity greater than unity if submerged in boiling water under 15 pounds gage pressure. The weight of the saturated balsa is up to 10 times its dry weight. Centrifuge tests have shown a water loss of less than 2 percent after 5 minutes at 5 g's. Most of this loss was probably due to evaporation rather than centrifugal-force effects. Heat-transfer tests in an apparatus that simulated the structure shown in figure 10 revealed that temperatures could be adequately controlled with saturated balsa as the heat sink. For a flight about 1 hour long with about 40 minutes of aerodynamic heating at the conditions considered in determining the heat load for figure 1 (18,000 ft/sec at an altitude of about 180,000 ft), the required balsa thickness would vary from about 0.1 to 0.3 inch, depending on the surface equilibrium temperature, which is a function of distance from the leading edge.

Certain difficulties are involved in the use of saturated balsa as a heat sink. It tends to distort upon drying; if exposed to air under the heating conditions encountered in this application it will char after drying; and resaturation may be difficult within the aircraft because it would require circulating pressurized hot water throughout the entire heat-sink area. Because of these difficulties balsa is probably not the best carrier for water in a built-in heat sink, and some other material may be better. This preliminary study has shown, however, that cooling with a built-in heat sink is feasible and that materials can be found that can absorb adequate amounts of water and not lose this water under acceleration forces up to at least 5 g's.

Weight Effects

In order to determine the optimum material to use in the main structure of the aircraft, it is necessary to know the effect of structural temperature on the weight of the cooling system. In general, structures designed for higher temperatures will be heavier, and the cooling systems will be lighter; consequently, the weight of structure has to be balanced against the weight of the cooling system. Several weight comparisons are made for a double-wall insulated cooling system in figure 11 for hydrogen and water as heat sinks. In this figure the weight of the cooling system is given per square foot of aircraft surface, and it is taken as the weight of 3-pound-per-cubic-foot Thermoflex insulation plus the weight of heat-sink material required to absorb the heat that would have to be removed during 40 minutes of aerodynamic heating. The cooling-system weights obtained by this analysis are probably optimistic. No tank or ducting weight was included for hydrogen, and it was assumed that the hydrogen could be heated to the same temperature as the structure. This assumption gives the maximum heat capacity possible. For water a 10-percent dead weight was included in the calculations to account for balsa wood used to hold the water. Experience may show, however, that another 5- or 10-percent increase in weight may be necessary because of difficulty in obtaining complete drying of the balsa during flight.

Figure 11(a) shows the effect of insulation thickness for an outer-skin temperature of 1800° F. As would be expected, the weight is less with hydrogen than with water as the heat sink because of the greater heat capacity of hydrogen. In addition, water could not be used for structure temperatures higher than about 212° F for reasonable internal wing and fuselage pressures because water absorbs its heat by evaporation. With hydrogen, the higher the structure temperature, the more heat can be absorbed because of higher temperature rises to the heat sink. This is doubly effective because the heat flux is decreased owing to smaller temperature gradients through the insulation at higher structure temperatures. Figure 11(a) shows that it would be detrimental to increase the insulation thickness from 1 inch to 2 inches for hydrogen, because the increased weight of insulation more than outweighs the reduction in hydrogen required for cooling. If water is used as the heat sink, however, it would be advantageous to increase the insulation thickness, if possible, to reduce weight.

Figure 11(b) shows how insulation thickness could be decreased in the aircraft as structural temperature is increased and a constant heat load is maintained. The weight saving shown as structure temperature is increased and insulating thickness is decreased is due primarily to added heat capacity in the hydrogen at high structure temperatures. The effect of insulation weight reduction is very small. Comparison of figures 11(a) and (b) shows that 1 inch of insulation thickness approaches an optimum for hydrogen as the heat sink. For a given structure temperature the coolant-system weights for 1 inch of insulation are slightly less than those for thinner insulation, but the effect of thickness is quite small. This indicates that, if hydrogen is used as the heat sink, there can be considerable freedom in the choice of insulation thickness.

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Figure 11(c) shows the effect of skin temperature on coolant-system weight. The equilibrium temperatures of the skin will vary with distance from the leading edge, flight speed, and altitude. The skin temperature has a significant effect on the weight of water required for a heat sink but only a small effect if hydrogen is the heat sink.

Increasing the structure temperature from 212° up to 1200° F could result in a maximum reduction in hydrogen cooling-system weight of about 1/2 pound per square foot of surface area (fig. 11). An analysis of the structure is needed to determine whether this weight saving would warrant building the structure of high-temperature materials. In addition, from 0.3 to 1.3 pounds per square foot of aircraft surface can be saved by using a hydrogen cooling system rather than a water cooling system, depending on structure temperature and insulation thickness. This weight saving has to be balanced against the volume that would be required for storing hydrogen. Essentially no useful volume is taken by the water if it is in balsa imbedded within the aircraft structure.

CONCLUSIONS

From this preliminary investigation, the following conclusions can be drawn:

1. Several heat sinks are probably feasible for hypersonic aircraft. Water and liquid hydrogen appear to be most practical, but sodium also looks promising. At the present time lithium and lithium hydride do not appear to be suitable as heat sinks because of difficulties in handling molten lithium.

2. It appears that storage and circulation of hydrogen in an aircraft need not be a hazard; therefore, hydrogen would be better as a heat sink than helium, which would have similar handling and storage problems but less heat capacity than hydrogen.

3. Cooling or temperature-reduction devices that were studied and that do not appear to be practical at the present time in high-level regions are: (1) use of the upper wing surface as a heat sink, because of weight and complication of ducting and pumps required, (2) low-thermal-conductivity coatings, because known coatings that may be applicable have such low emissivities that they do not reduce the heat flux to the coolant, and (3) transpiration cooling, because coolant-flow distributions that are presently obtainable in regions where there are large pressure gradients are so far from ideal that coolant requirements may be no smaller than for convection cooling.

4. Uncooled surfaces made of silicon carbide, impregnated graphite, or tungsten with a silicide coating appear to be feasible in the high-level regions of the airplane, so that high-level cooling could possibly be eliminated.

5. If it is not desirable to use uncooled surfaces in the high-level regions, cooling could be accomplished with hydrogen, water, sodium, or a sodium-potassium mixture. A counterflow cooling system should be used, particularly with hydrogen, to reduce temperature variations and prevent overcooling in some areas. Hydrogen has the advantage that pumps and heat exchangers would not be required.

6. Practically any coolant could be used for convection cooling in low-level regions, but water and hydrogen are particularly promising because they are also good heat sinks. A cooling-system weight analysis showed that increasing the structure temperature from 212° to 1200° F could result in a maximum reduction in hydrogen cooling-system weight of about 1/2 pound per square foot of surface area. In addition, the hydrogen system can be from 0.3 to 1.3 pounds per square foot lighter than the water system, depending on structure temperature and insulation thickness.

7. The reduced weight of a hydrogen cooling system has to be balanced against the volume required for hydrogen storage. It appears possible to eliminate the necessity of taking useful volume from the aircraft for heat-sink storage by imbedding balsa saturated with water within the aircraft structure for low-level cooling and by using uncooled surfaces of high-temperature materials in the high-level regions.

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Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, January 7, 1958

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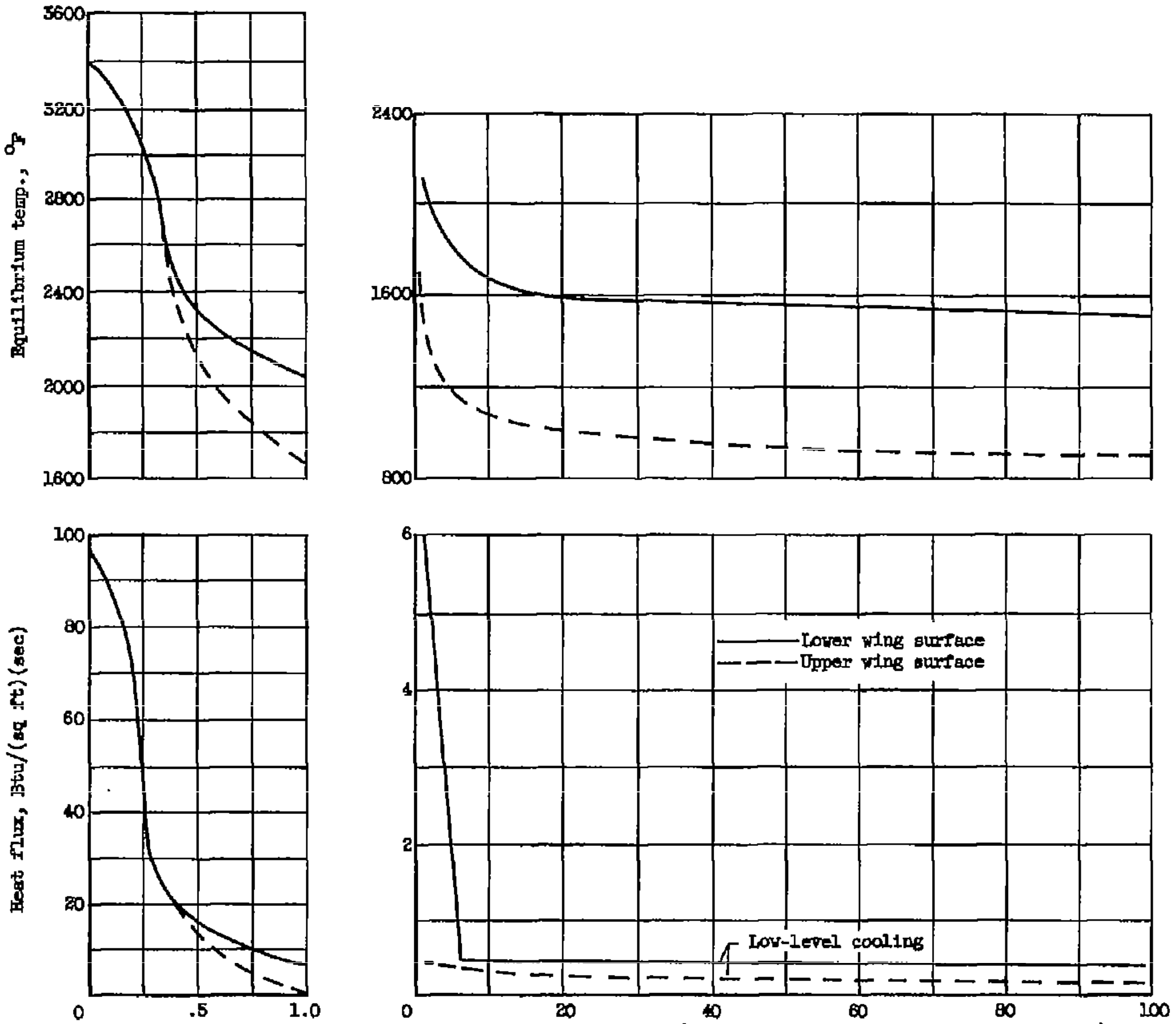

Figure 1. - Variation of equilibrium temperature and heat flux on leading-edge portion of wing of a hypersonic aircraft. Flight speed, 18,000 feet per second; altitude, 180,000 feet; wing loading, 20 pounds per square foot; angle of attack, 5°; sweep angle, 78°; leading-edge radius, 0.25 inch.

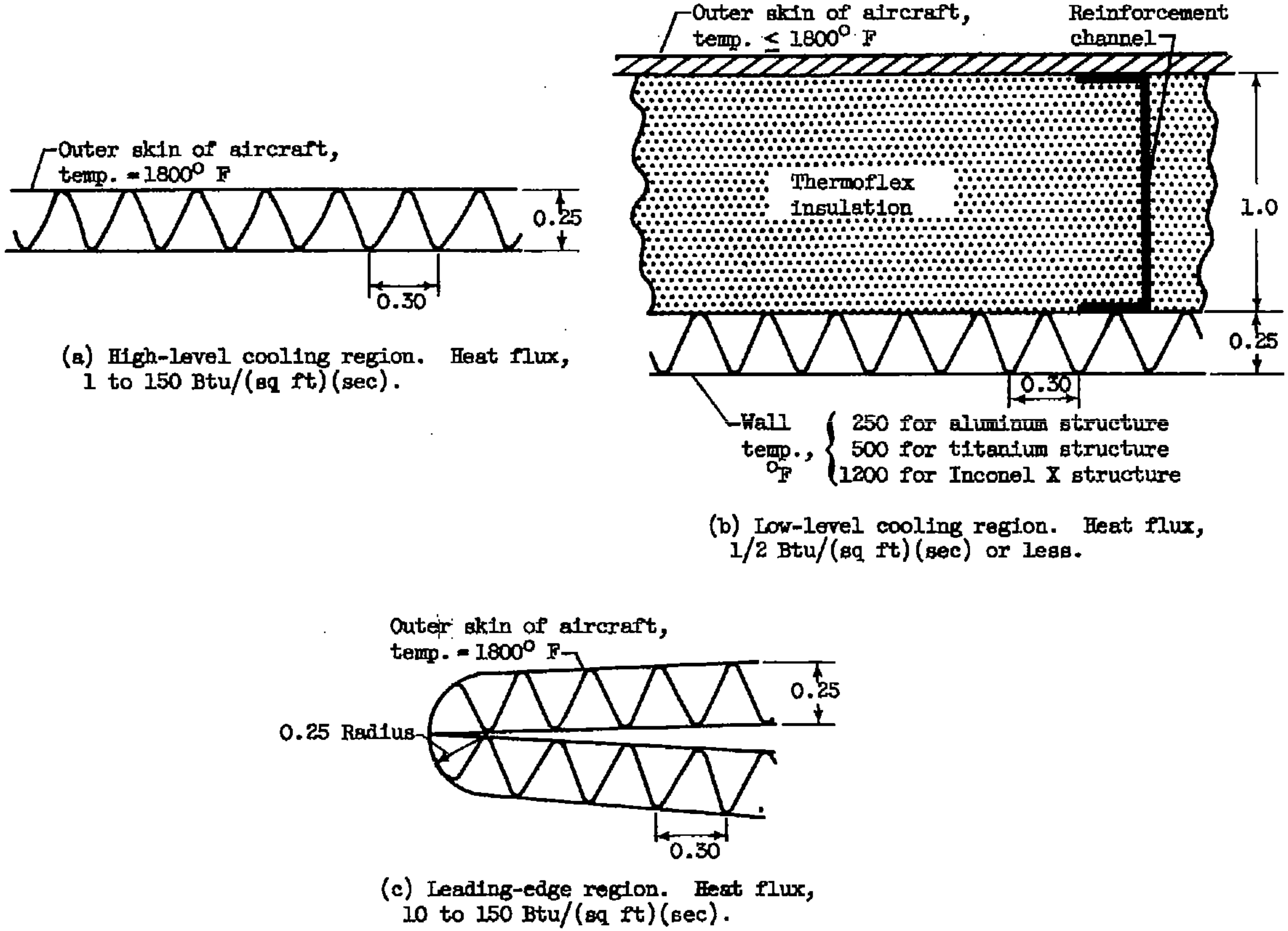

Figure 2. - Structural geometries, temperatures, and heat fluxes considered in cooling analysis (dimensions in inches).

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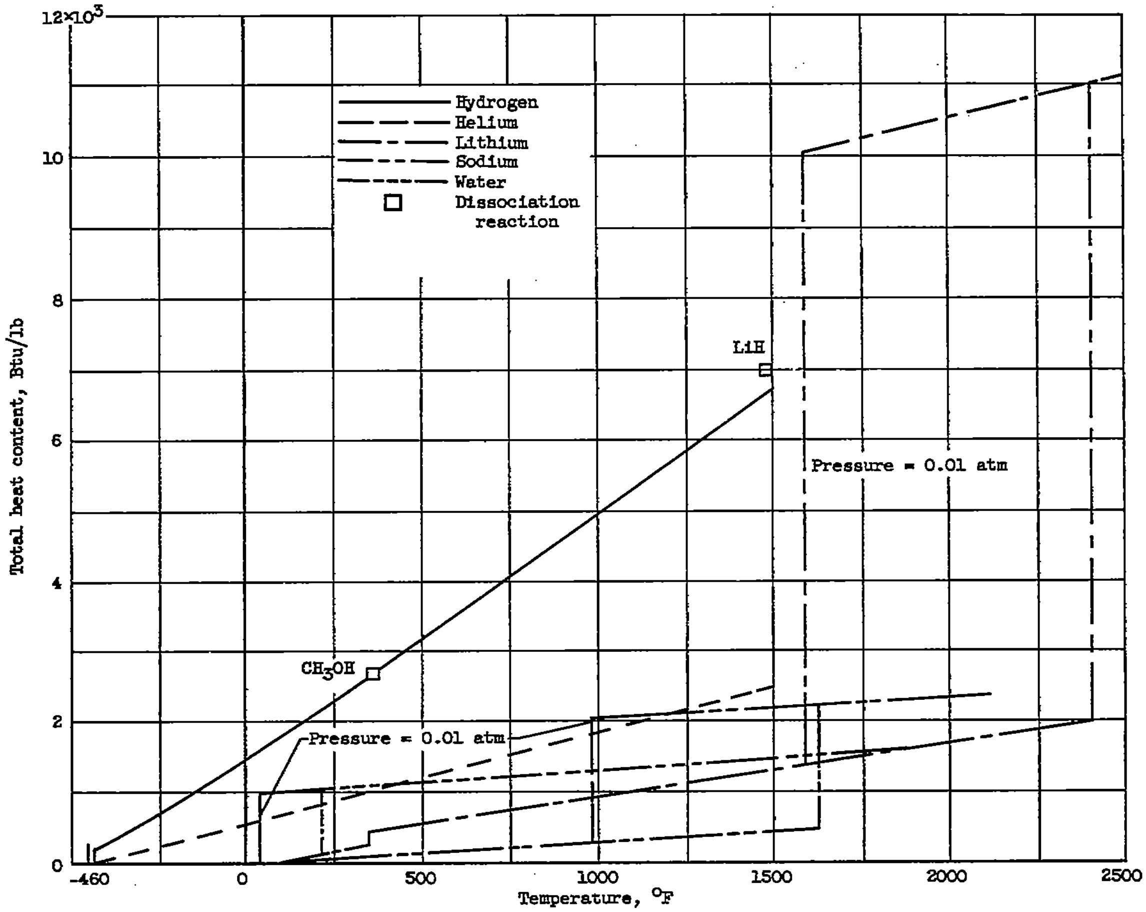
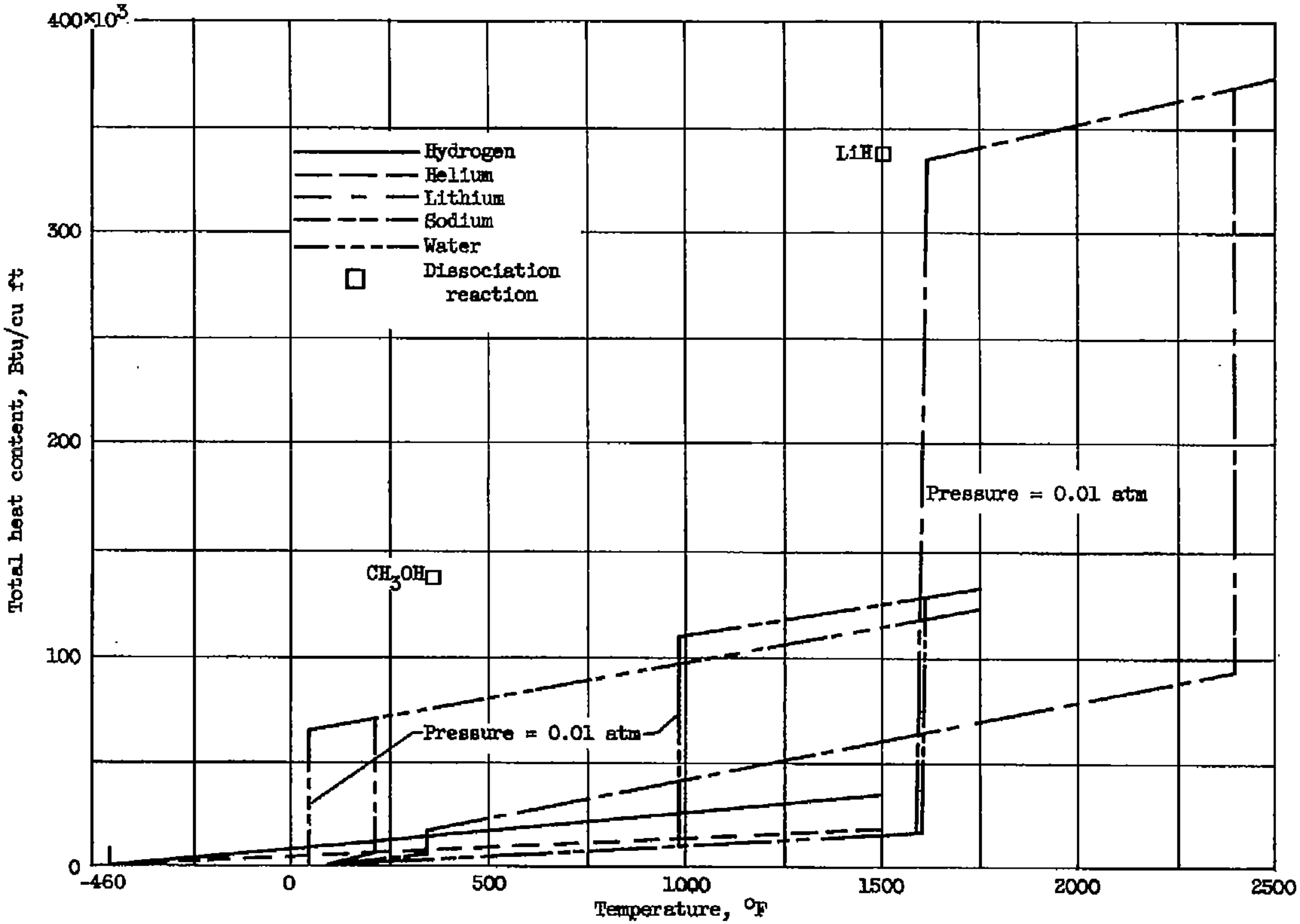


(a) Heat content on a density basis.

Figure 3. - Total heat content of several materials for a range of temperatures. Heat content assumed zero at storage temperature; pressure, 1.0 atmosphere unless noted otherwise.



(b) Heat content on a volume basis.

Figure 3. - Concluded. Total heat content of several materials for a range of temperatures. Heat content assumed zero at storage temperature; pressure, 1.0 atmosphere unless noted otherwise.

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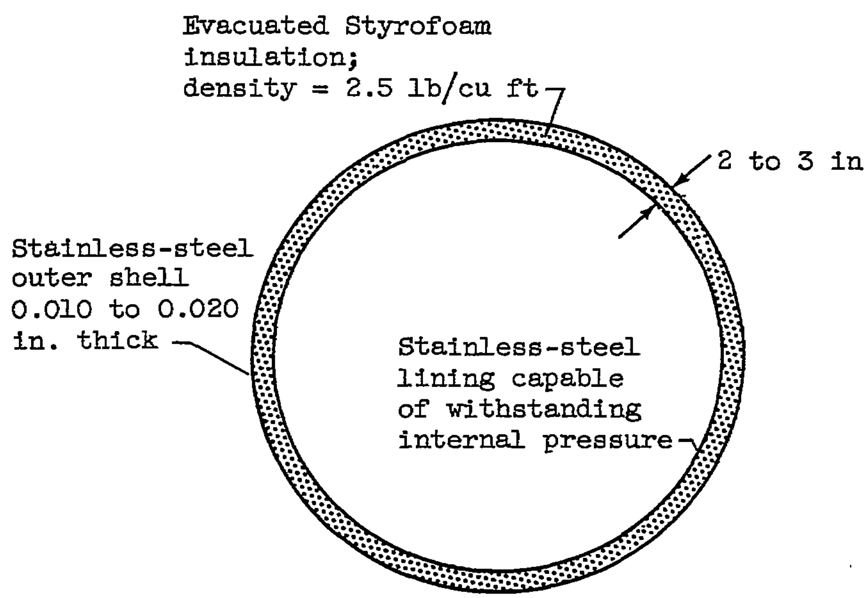
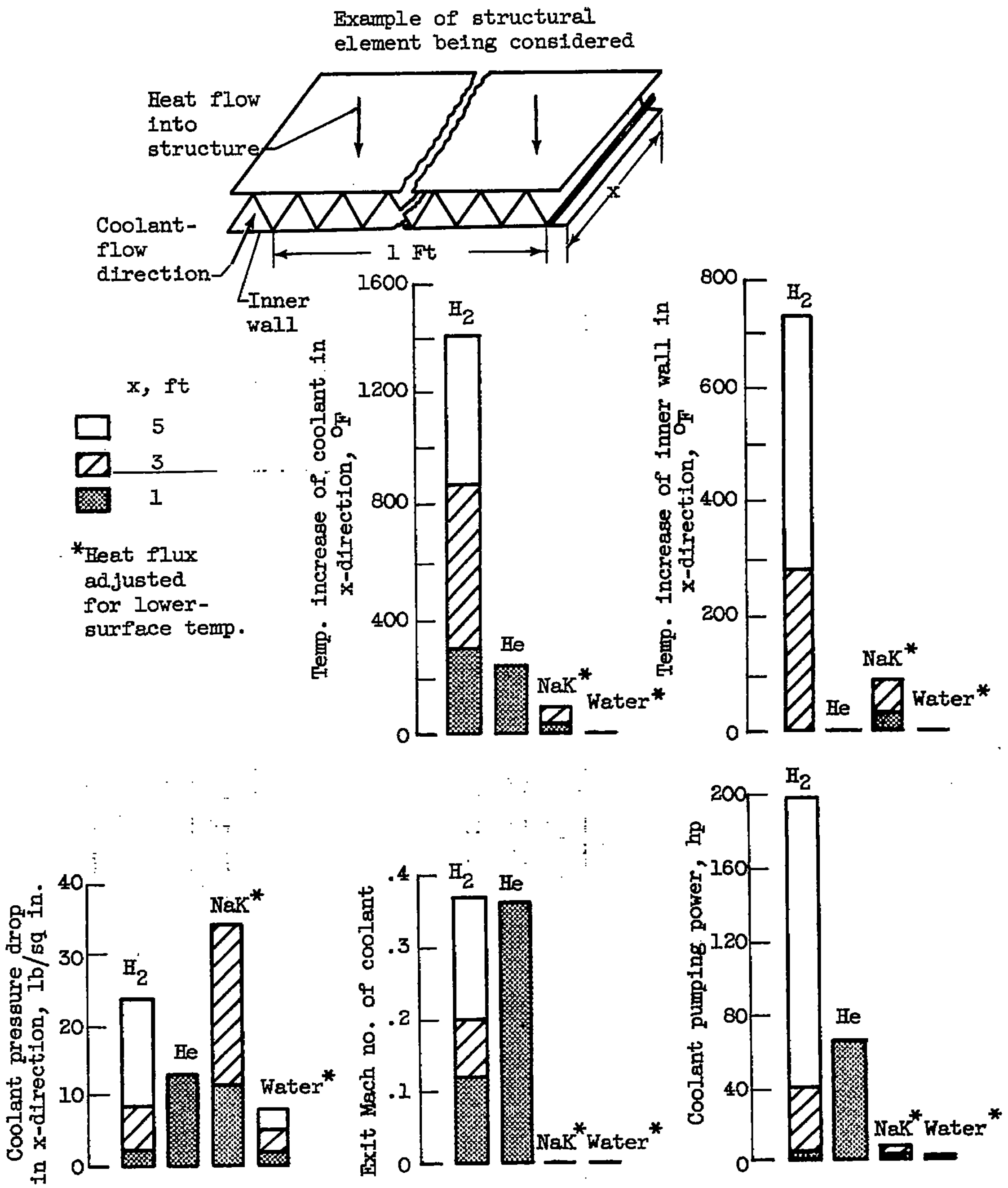


Figure 4. - Possible configuration for an insulated, vacuum-jacketed storage tank for liquid helium or hydrogen.



(a) Heat flux, 150 Btu/(sq ft)(sec).

Figure 5. - Comparison of several coolants.

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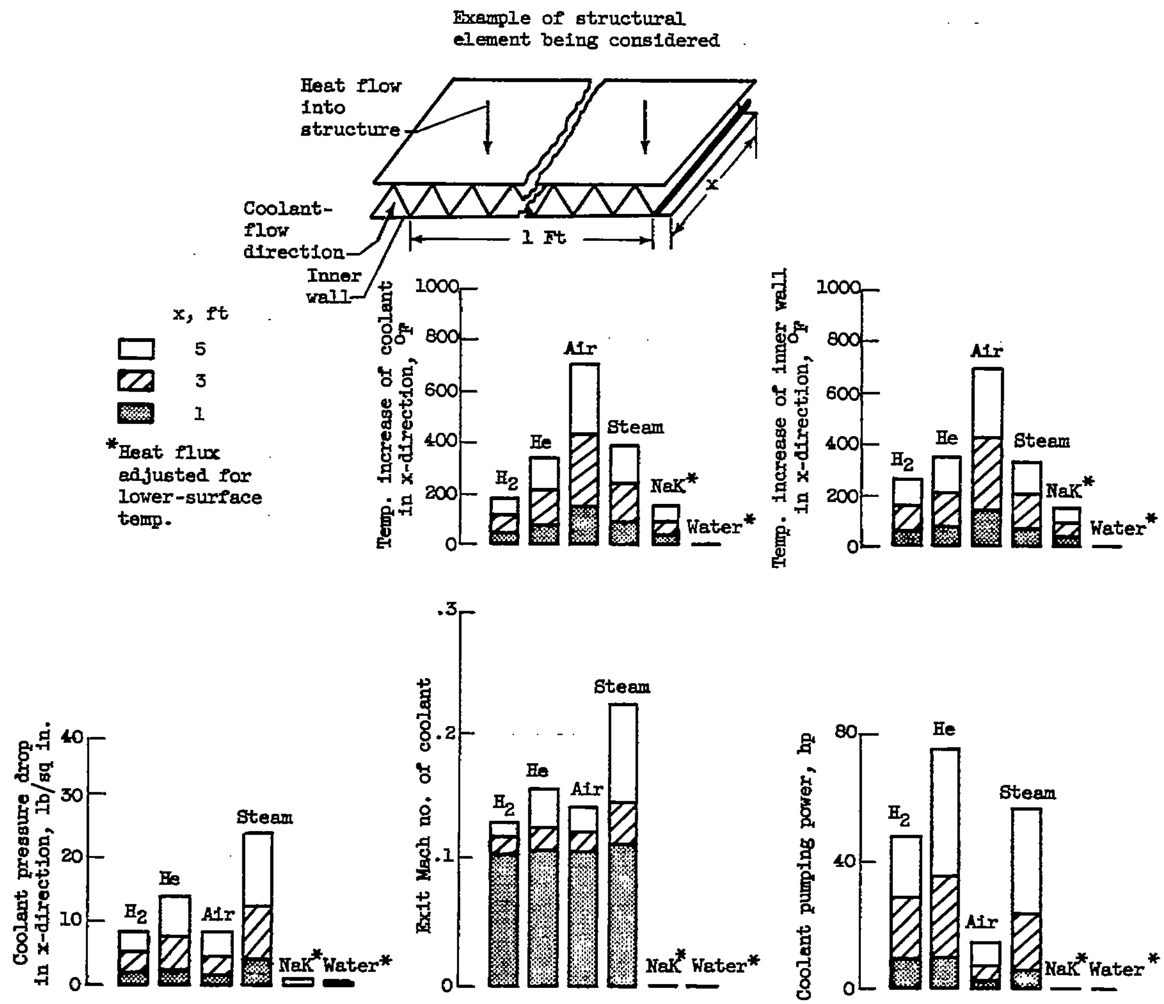

(b) Heat flux, 10 Btu/(sq ft)(sec).

Figure 5. - Concluded. Comparison of several coolants.

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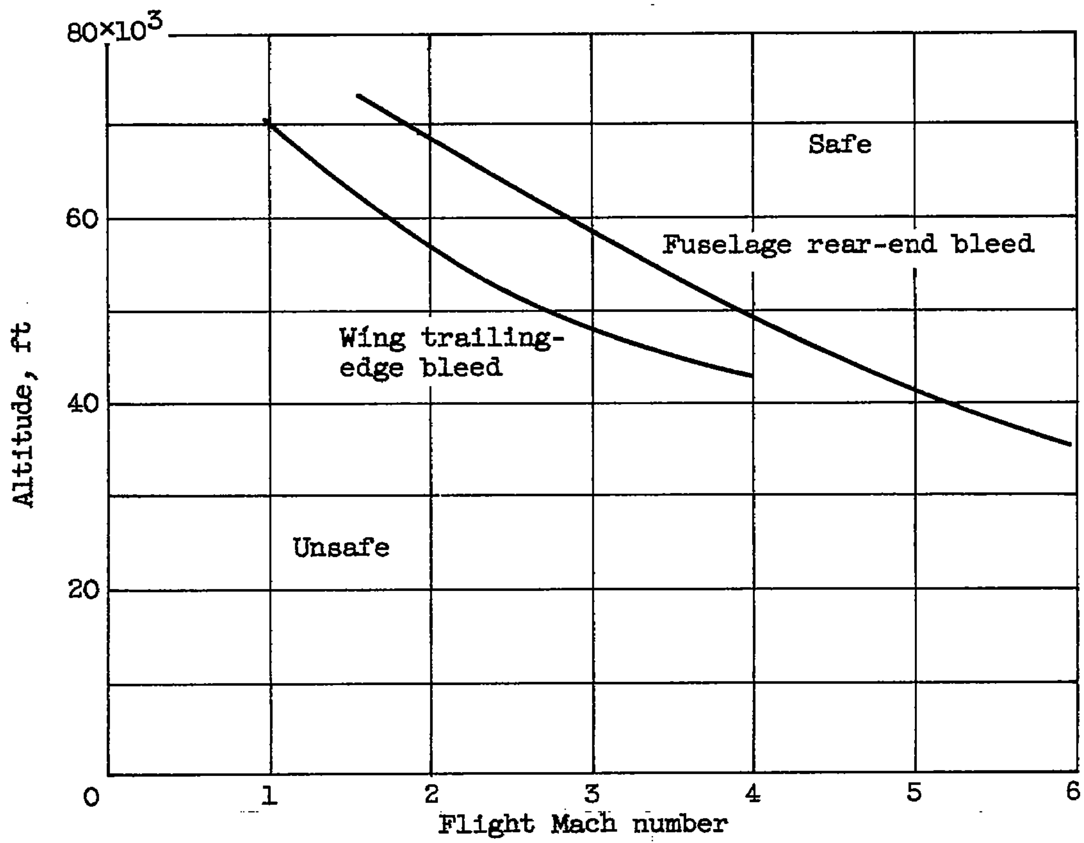

Figure 6. - Minimum altitude for purging cooling system for hydrogen detonation pressure less than 2 atmospheres.

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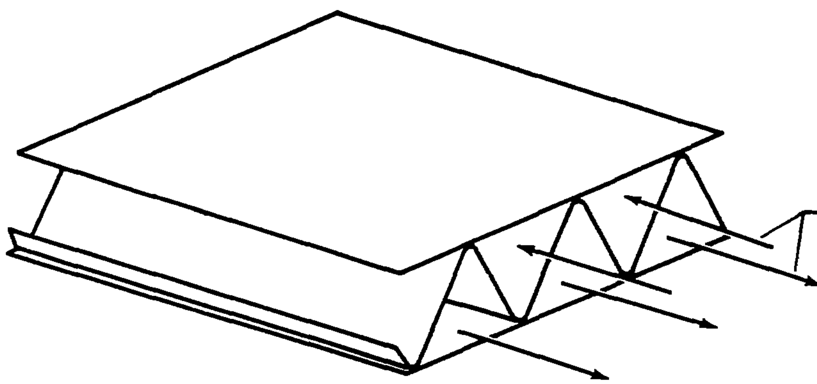

Figure 7. - Example of counterflow cooling of a corrugated structure.

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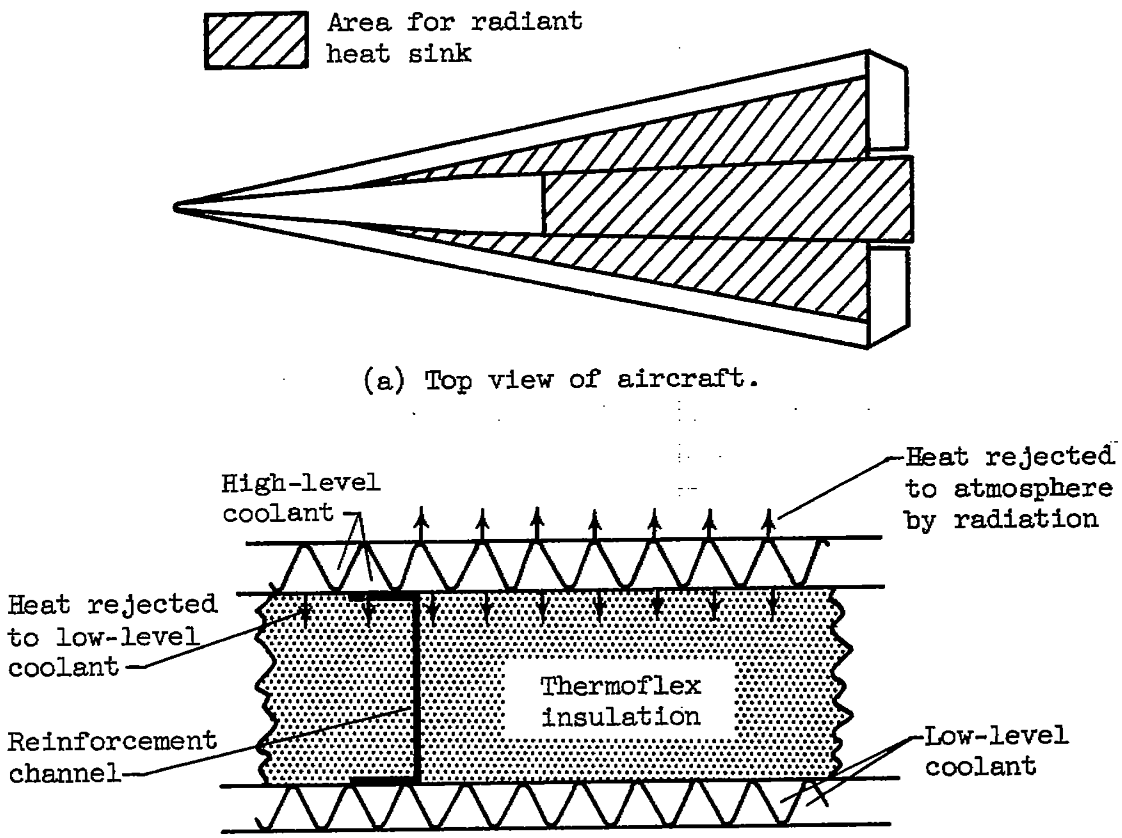

(a) Top view of aircraft.

(b) Cross section of structure in radiant heat-sink area.

Figure 8. - Example of using the upper surface of hypersonic aircraft as a radiant heat sink for high-level coolant.

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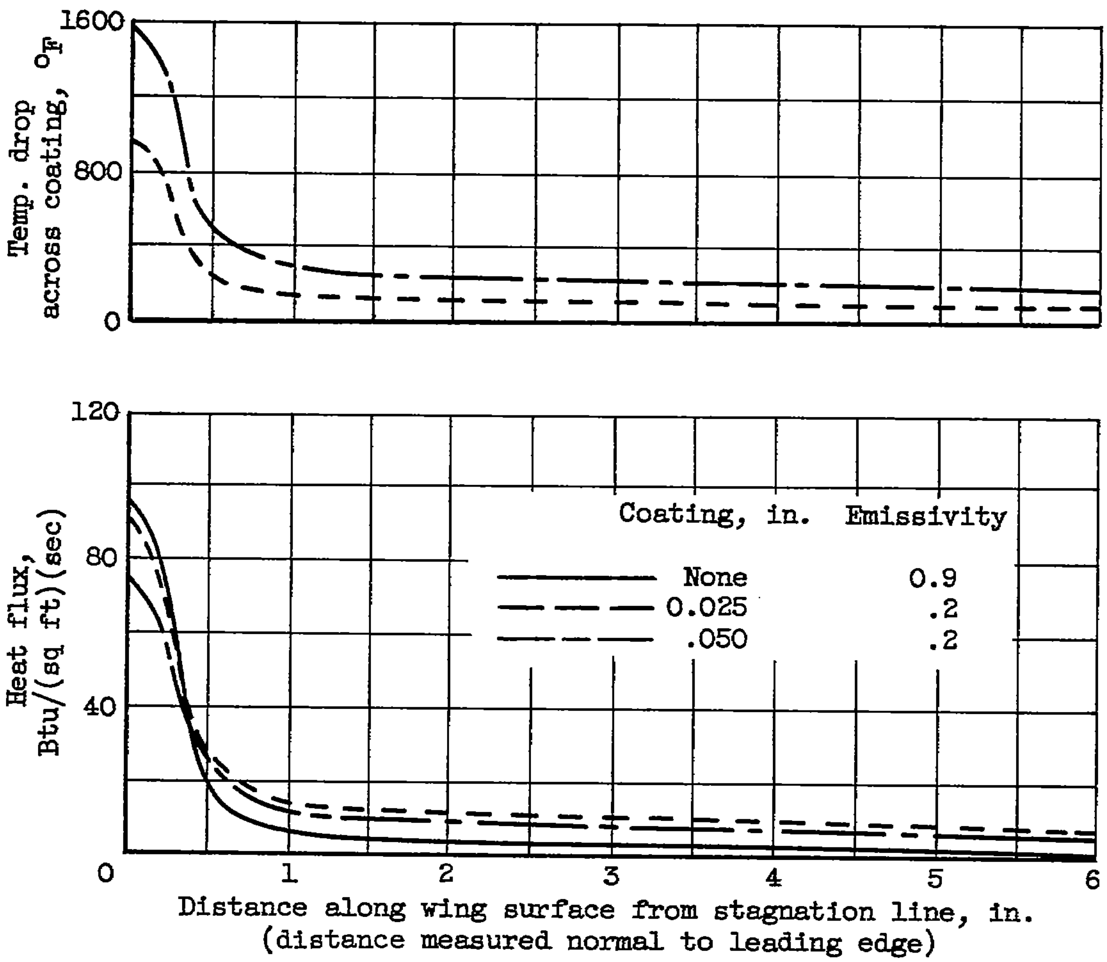

Figure 9. - Variation of temperature drop and heat flux through zirconia coatings applied to leading-edge region of wing of a hypersonic aircraft. Flight speed, 18,000 feet per second; altitude, 180,000 feet; wing loading, 20 pounds per square foot; angle of attack, 5°; sweep angle, 78°; leading-edge radius, 0.25 inch.

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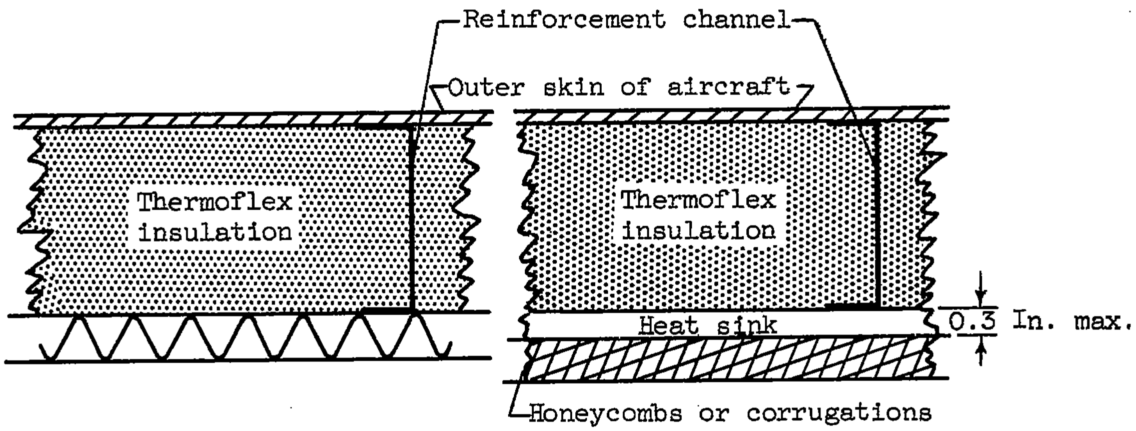
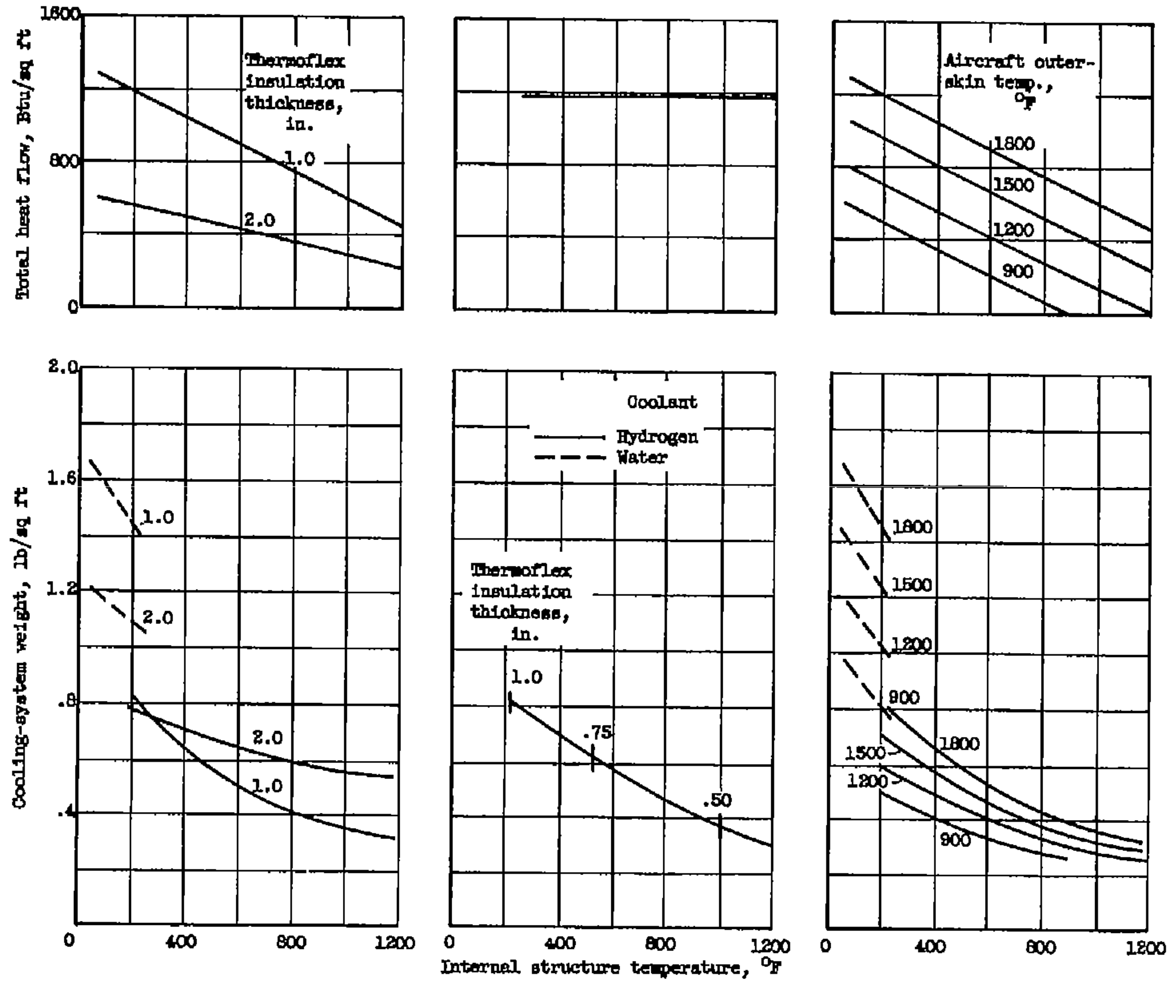


(a) Convection-cooled structure.

(b) Heat-sink-cooled structure.

Figure 10. - Possible low-level cooling methods.



(a) Effect of insulation thickness on total heat flow and cooling-system weight (aircraft outer-skin temp., 1800° F).

(b) Effect of insulation thickness required to give constant total heat flow on cooling-system weight (aircraft outer-skin temp., 1800° F).

(c) Effect of aircraft outer-skin temperature on total heat flow and cooling-system weight for 1 inch of Thermoflex insulation.

Figure 11. - Comparison of hydrogen- and water-cooling systems for range of internal structure temperatures.